NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE

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WAKE MEASUREMENTS BEHIND A WING SECTION OF A FIGHTER AIRPLANE IN FAST DIVES

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SUMMARY

Wake measurements made in a vertical plane behind a wing section of a fighter airplane are presented for a range of Mach number up to 0.78. Since evidences of reverse flow were found in a large part of the surveys — possibly because of interference of the rake support — the computed profile—drag coefficients are considered to be only qualitative.

The results showed that the large increase in drag coefficient beyond the critical Mach number indicated by wind—tunnel tests was also obtained under flight conditions and that the wake width was extended sharply when shock was encountered. The wake extension occurred first at the upper surface since the highest local velocity was obtained on that surface. The large increase in drag coefficient for the wing section tested did not occur until after the critical Mach number had been exceeded by approximately 0.05. Comparison of the profile-drag measurements with total airplane drag measurements showed that the large increases in drag in both cases started to occur at the same value of Mach number.

The results further indicated that wake measurements made in three—dimensional flow after shock had occurred cannot, in general, be interpreted in terms of section profile—drag coefficient because of the existence of the strong lateral flow indicated by tuft behavior in the dead-air region behind the shock.

INTRODUCTION

During dive test on the fighter airplane tested, measurements of the profile drag through and beyond critical speed were required in order to obtain data for comparison with similar measurements made in a wind tunnel. The airplane was accordingly equipped with static-pressure and total-pressure survey rakes mounted behind the left wing at about semispan location. Several dives were made with this equipment installed, and measurements were taken at Mach numbers between 0.31 and 0.78.

A long survey rake was necessary for the purpose of obtaining measurements of the pressure losses due to shock at appreciable distances above and below the plane of the wing. Structural difficulties imposed by the air forces acting on this long rake at high diving speeds required that the structural elements of the rake and of the supporting member be sturdy and that the whole assembly be mounted quite near the trailing edge of the wing.

The pressure surveys made with this rake equipment showed evidences of reverse flow at the center of the turbulent wake, possibly a consequence of the design conditions described. The profile—drag data obtained are, therefore, of only qualitative value.

The results show, however, the value of the Mach number at which the expected large increase in the drag coefficient occurs. The pressure losses behind the shock outside the turbulent wake were also correctly measured, and the width and position of the turbulent wake at the rake location as functions of the Mach number were correctly determined.

The profile—drag curve given in the present paper, although of only qualitative value, is compared with the airplane over-all drag curve obtained to show that the large increase in drag occurs at the same value of Mach number in both cases and further to show the apparently greater increase in the profile drag than in the over-all drag with increasing Mach number. Observation of the flow pattern as disclosed by the behavior of wool yarn tufts secured to the upper surface of the wing indicated that it was possible for this greater increase to occur and also indicated that even with a favorable rake installation profile—drag measurements are not quantitatively reliable in three-dimensional flow beyond the critical speed where lateral flow exists in the dead-air region behind the shock.

SYMBOLS

c_n section normal-force coefficent

M Mach number

c chord of wing section forward of rake

c_r chord of rake

t_r thickness of rake

S static-pressure tube

P pressure coefficient

c_{d_o} section profile-drag coefficient

y/c distance along rake from trailing edge in percent of chord.

APPARATUS

Airplane.- A front view of the fighter airplane tested is shown as figure 1. During the tests the airplane was in service condition and was coated with camouflage paint. No attempt was made to finish the wing to an aerodynamically smooth condition. The point of transition from laminar to turbulent flow along the chord was not measured. The machine-gun openings in the leading edge of the wing and the lower edge of the ammunition door were taped.

Rake and wing section.- The wake was surveyed by means of a rake mounted on the flap of the left wing at a distance of 51.3 percent of the semispan from the plane of symmetry. Details of the installation may be obtained from examination of figures 1 to 3. The wing is a modified NACA 44-series low-drag wing, and the section at the rake location is approximately 14 percent thick. The measured ordinates of the profile are given in the tables in figure 3. The simultaneous measurement of the wing pressures and the wake survey limited the number of rake tubes to a total of 30, of which 24 were total-pressure tubes and six were static-pressure tubes. Both the static-pressure and total-pressure tubes were of brass tubing and had outside and inside diameters of 0.188 inch and 0.124 inch, respectively. The total-pressure tubes extended $3\frac{1}{2}$ inches forward of the vertical supporting structure. The static-pressure tubes were offset about 1 inch from the plane of the support, and the static-pressure holes were located about 5 inches forward of the rake, where calculations indicated that interference velocity due to the support would be small.

Although the wind-tunnel tests (reference 1) had previously shown that wake losses could extend 1 chord above or below the wing, ground clearances during landing with flaps extended and structural limitations prevented the installation of a rake long enough to measure such wake losses. The rake installed extended 22.6 percent of chord above and 19.9 percent of the chord below the wing. In addition, rows of tufts were placed on both sides of the survey station to determine the air-flow behavior over the wing.

Instrumentation.- Measurements of the following quantities were obtained by standard NACA recording instruments synchronized by a timer: indicated airspeed, pressure altitude, wing surface pressures over a section forward of the rake, total and static pressures across the rake, and normal acceleration.

The pressure system used in obtaining the pressure measurements is shown in figure 4. The total pressures at the rake were measured with respect to the pressure at a tube extending forward of the leading edge of the left wing. (See fig. 3.) The static pressures at the rake and the pressures on the wing surface were measured with respect to the static pressure at the airspeed head; the static pressures were in turn corrected for position error.

METHOD AND RESULTS

Measurements were taken during parts of power-off dives started at the airplane ceiling (approx. 30,000 ft) and during the subsequent recoveries, thus covering a range of lift coefficient and Mach number. In general, results at values of M higher then 0.07 were obtained at approximately 20,000 feet, whereas data at lower Mach numbers were obtained at pressure altitudes near 25,000 feet.

Typical wake surveys obtained at various Mach number and the corresponding chordwise pressure distributions are shown in figures 5 and 6, respectively. When these results were obtained, the value of section normal-force coefficient c_n was between 0.1 and 0.2. The critical Mach number of the wing section forward of the rake was determined from measured pressures on the section that corresponded to the local sonic speed; for section normal-force coefficients of 0.2 the critical Mach number is approximately 0.67.

Approximately 60 surveys of the type shown in figure 5 were integrated to obtain qualitative values of profile drag throughout the Mach number range. Results of these integrations are given in figure 7 for a range of Mach number from 0.31 to 0.78. The symbols with tails attached represent the results from the rake surveys shown in figure 5.

At low Mach numbers the total pressures at the center of the wake are very near the prevailing static pressures and at high Mach numbers are less than the static pressure. (See fig. 5.) Throughout the entire Mach number range therefore some degree of reverse flow is indicated near the center of the wake. In the evaluation of the data for conditions in which the total-pressure reading was less than that of the static pressure, the two pressures were assumed to be equal. Under these conditions the integration of wake surveys cannot be said to yield a true measure of the profile drag, and the degree of error cannot be established without extensive additional experiments. The width and the location of the turbulent wake are nevertheless established as well as the Mach number at which the large increase in drag occurs.

DISCUSSION

The degree of accuracy of the flight data is, in general, more dependent upon the limitations of the rake design and installations than upon the instruments. An analysis of all the possible causes of error indicated that the profile-drag coefficient would be in error by not more than ± 5 percent if only instrument errors and personal errors affected the accuracy. As previously indicated, however, anomalous flow conditions existed in the region of the rake, so that the over-all degree of error cannot be estimated.

From the typical diagrams of figure 5 the wake at supercritical velocities can be considered as composed of two parts: the center wake due to skin friction and separation losses, which contributes almost all the drag at the lower Mach numbers, and the shock wake, identified by the total-pressure loss on either side of the center wake and attributed to the shock that extends from the boundary layer. At Mach numbers considerably higher than the critical Mach number, it would be expected that the shock losses would account for a large part of the drag and that the shock losses would increase in magnitude along with an increase in the losses in the center wake.

The results given in figure 7 show that the profile-drag change is very small up to a Mach number approximately 0.05 greater than the critical Mach number for the wing section but that the profile-drag coefficient increases rapidly above this Mach number. The large increase in profile-drag coefficient is accompanied by an extension of the wake width. (See fig. 5.) The wake extension occurs first at the upper surface since the highest local velocity is obtained on that surface.

Figure 8 shows the faired profile-drag curve from figure 7 together with preliminary data from tests made in the Ames 16-foot high-speed tunnel on a $\frac{1}{3}$- scale model of the test airplane at the same spanwise station as used in the flight tests. Comparison of the two curves shows that the large increase in drag starts to occur at approximately the same Mach number and also that differences exist in the drag coefficients obtained from the two tests. Investigations of surface-condition effects on drag have shown profile-drag coefficients to be 0.003 to 0.005 higher for wings in service condition than for wings that were aerodynamically smooth. The differences shown in the present comparison are believed to be due mainly to surface conditions, since no attempt was made to smooth the camouflage paint on the airplane wing whereas the model wing was aerodynamically smooth. Also included in figure 8 is a general over-all drag curve for the airplane tested; this curve shows that the large increase in over-all drag occurs at approximately the same Mach number as the increase in profile drag from both wind-tunnel and flight tests.

Figure 9 shows the local pressure variation with Mach number for the chordwise station at which a marked change of flow first occurs as observed from tuft behavior. The Mach number (0.71) at which the abrupt change in pressure coefficient occurs is approximately the same Mach number at which the large increase in drag starts to occur.

Photographs taken of wool tufts installed on the upper surface of the wing showed that the flow conditions were such that a reliable measure of profile-drag variation would be difficult to obtain for this type of wing at supercritical speeds even with a rake installation adequate for ordinary drag measurements. Figure 10 shows photographs of the tuft behavior at a subcritical speed (M = 0.65), at about the speed at which the large increase in drag starts to occur (M = 0.73, and at a higher speed (M = 0.75. All photographs were obtained when the airplane lift coefficient was about 0.2. With the exception of a very small lateral flow over the landing flap toward the inboard sections the flow over the wing at the subcritical speed is steady and directed backward over the wing. At the intermediate speed slightly irregular flow together with slight inboard flow may be noted at approximately the 50-percent-chord station. At the highest speed the flow behind the shock is very turbulent and a pronounced inboard lateral flow is evident. With these flow conditions measurements obtained from any rake installation are not applicable to the evaluation of section profile drag.

CONCLUSIONS

Wake measurements have been made in a vertical plane behind a wing section of a fighter airplane for a range of Mach number up to 0.78; however, the computed profile-drag coefficients are considered to be only qualitative. The following conclusions can be made from analysis of these measurements:

1. At the wing section tested, the critical Mach number of the section was exceeded by 0.05 before large increases in the profile-drag coefficient occurred.

2. Large increases in drag coefficient beyond the critical Mach number such as shown by wind-tunnel tests of the fighter airplane model tested were also obtained under flight conditions and these increases started to occur at the same value of Mach number in both cases.

3. The large increase in profile-drag coefficient was accompanied by an extension of the wake width. The wake extension occurred first at the upper surface since the highest local velocity was obtained on that surface.

4. Wake measurements made in three-dimensional flow after shock had occurred cannot, in general, be interpreted in terms of section profile-drag coefficient because of pronounced lateral flow in the dead-air region behind the shock.

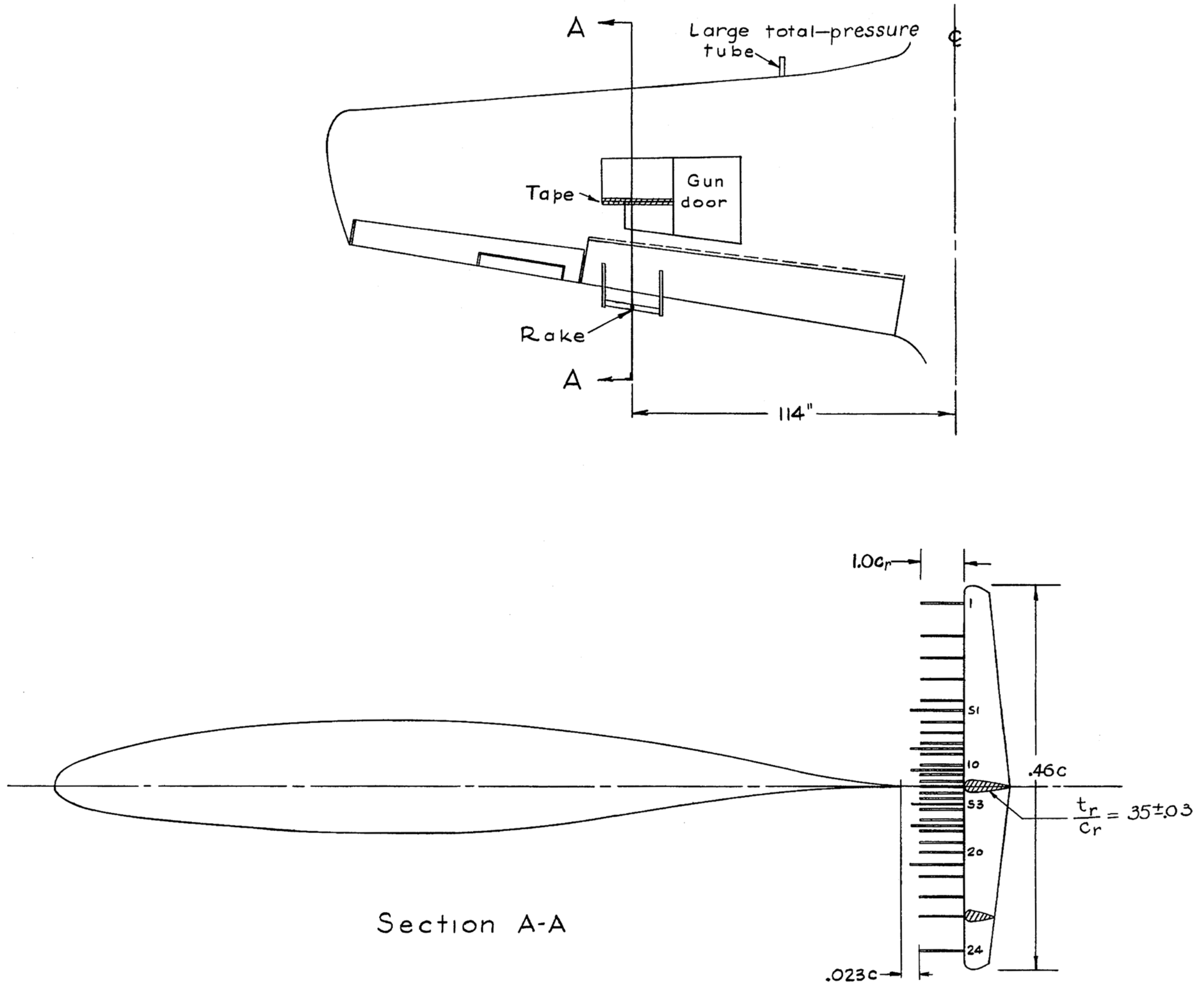
Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, VA., November 4, 1946

REFERENCE

1. Stack, John, Lindsey, W. F., and Littell, Robert E.: The Copressibility Burble and the Effect of Compressibility on Pressures and Forces Acting on an Airfoil. NACA Rep. No. 646, 1938.

Figure 1.- Front view of fighter airplane tested.

Figure 2.- Close-up of upper part of rake mounted on left wing of fighter airplane tested.



MEASURED WING-SECTION STATIONS AND ORDINATES
(In percent wing chord)

Station	0	1.25	2.50	5.00	7.50	10	15	20	25	30	35	40	50	60	70	80	90	95	100
Upper surface	0	1.86	2.62	3.65	4.45	5.11	6.11	6.86	7.44	7.90	8.15	8.21	7.82	6.83	5.36	3.50	1.46	0.66	0
Lower surface	0	1.65	2.08	2.99	3.53	3.96	4.61	5.13	5.46	5.18	5.92	5.90	5.58	4.50	3.07	1.55	0.48	0.04	0

RAKE TUBE LOCATIONS
(In percent wing chord)

Tube	1	2	3	4	5	S1	6	7	8	S2	9	10	S3	11	12
y/c	22.6	18.6	15.9	13.3	10.6	9.3	7.97	6.64	5.31	4.65	3.98	2.66	1.99	1.33	0.66
Tube	13	14	15	S4	16	17	S5	18	19	20	S6	21	22	23	24
y/c	0	-0.66	-1.33	-1.99	-2.66	-3.98	-4.65	-5.31	-6.64	-7.97	-9.3	-10.6	-13.3	-15.9	-19.9

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Figure 3. – Rake installation on left wing of fighter airplane tested.

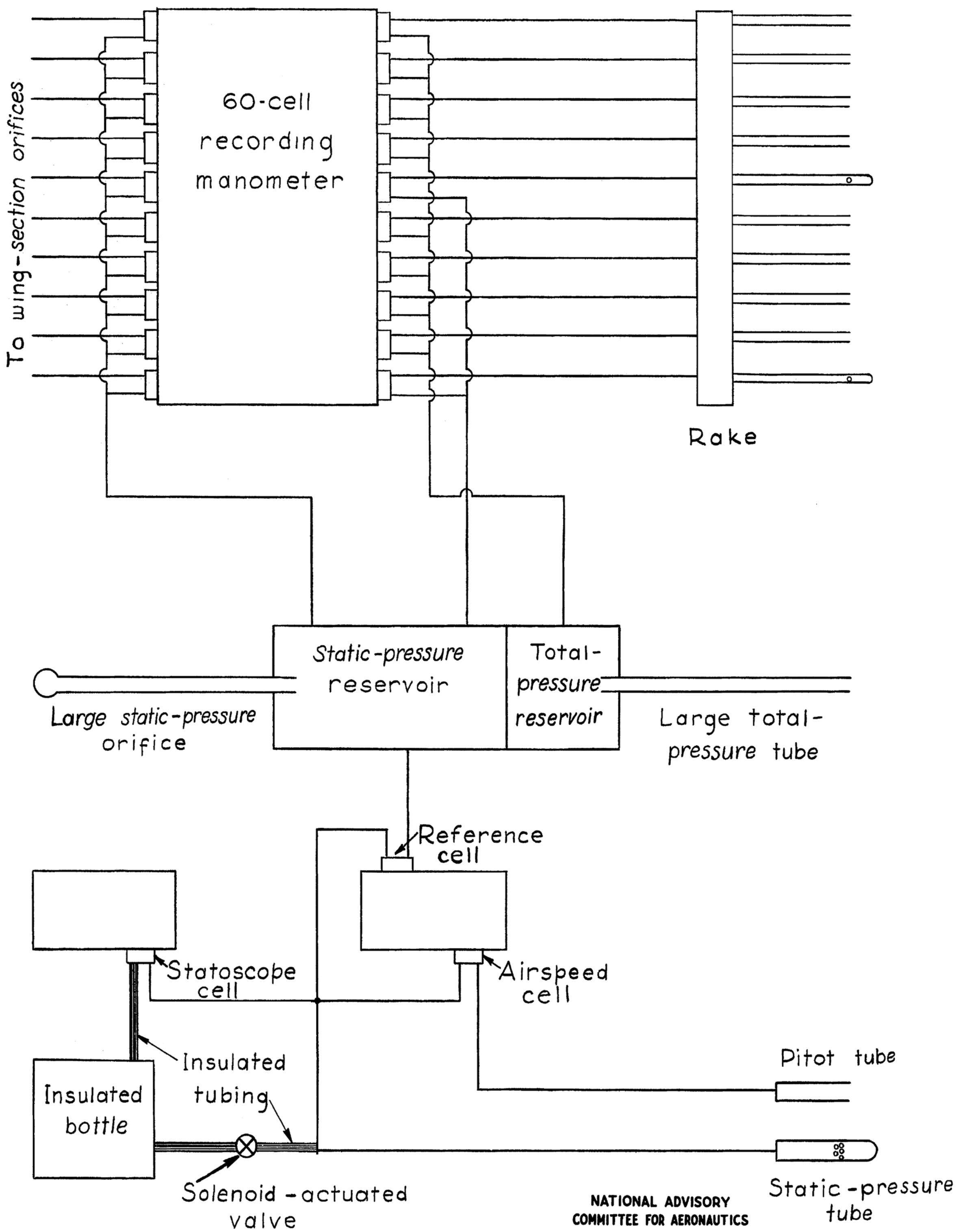


Figure 4.- Pressure system for determination of profile drag.

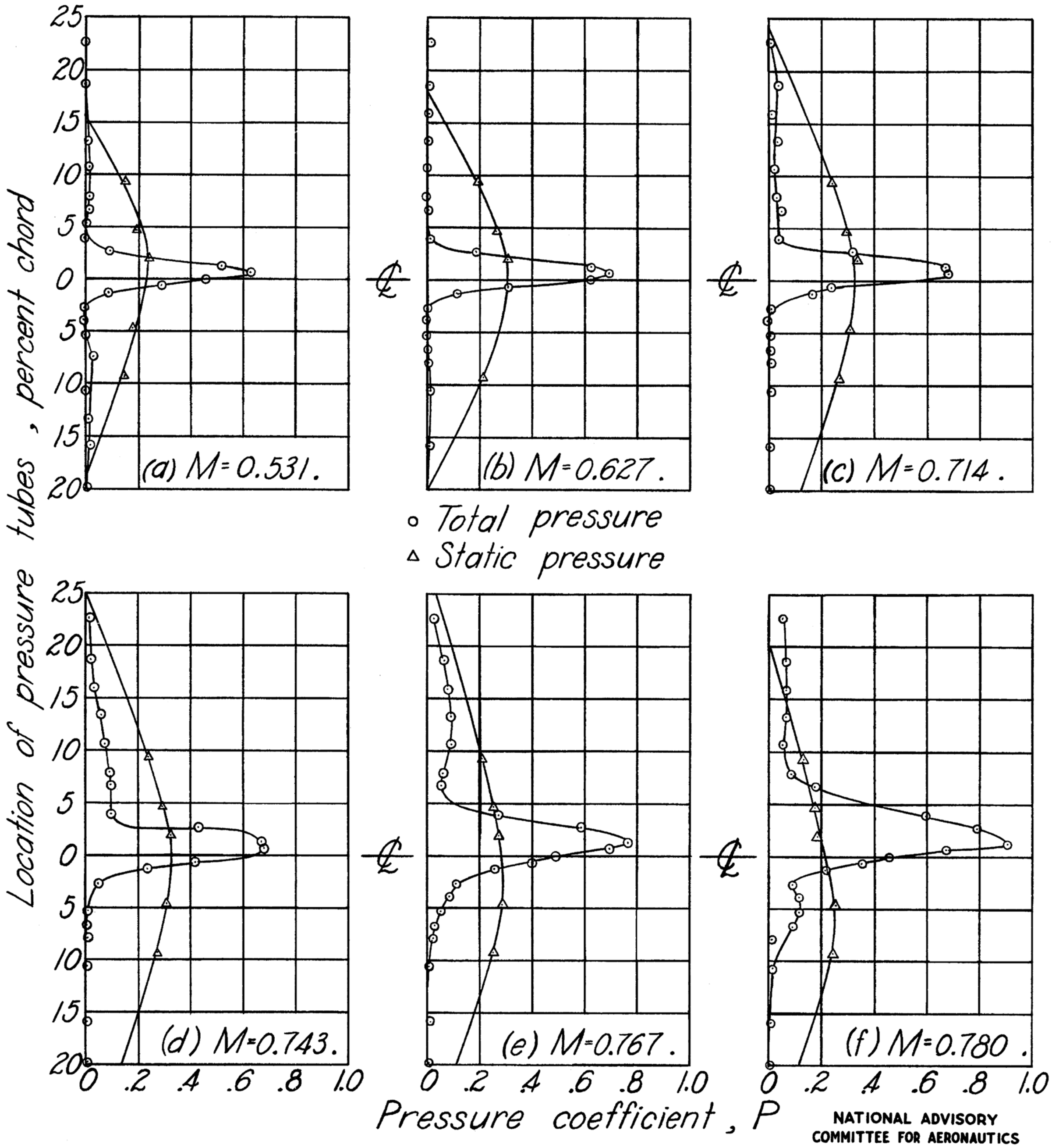


Figure 5.-Variation of wake shape with Mach number for $c_n < 0.2$.

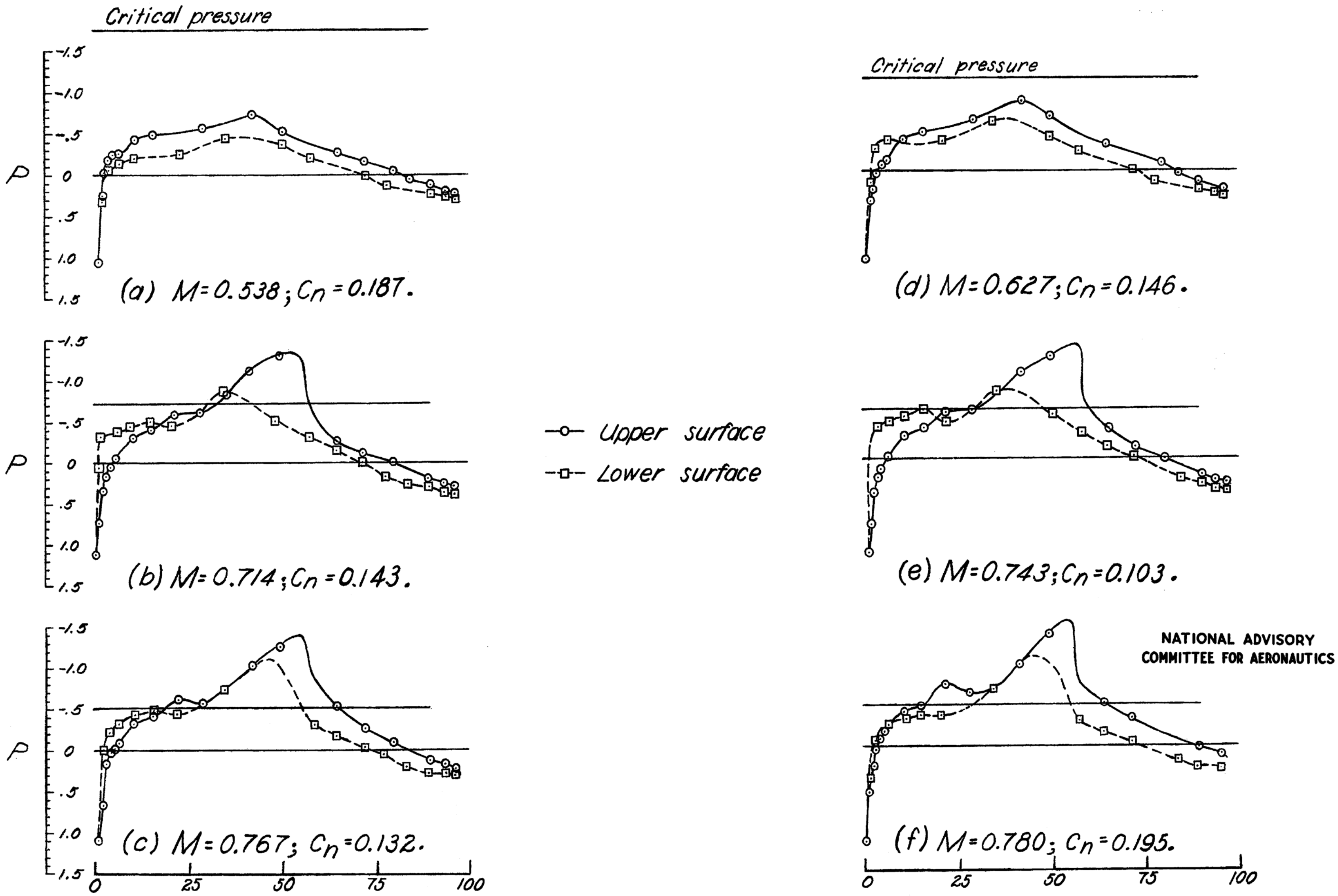


Figure 6.- Pressure distributions corresponding to wake distributions of figure 5. $0.1 < c_n < 0.2$.

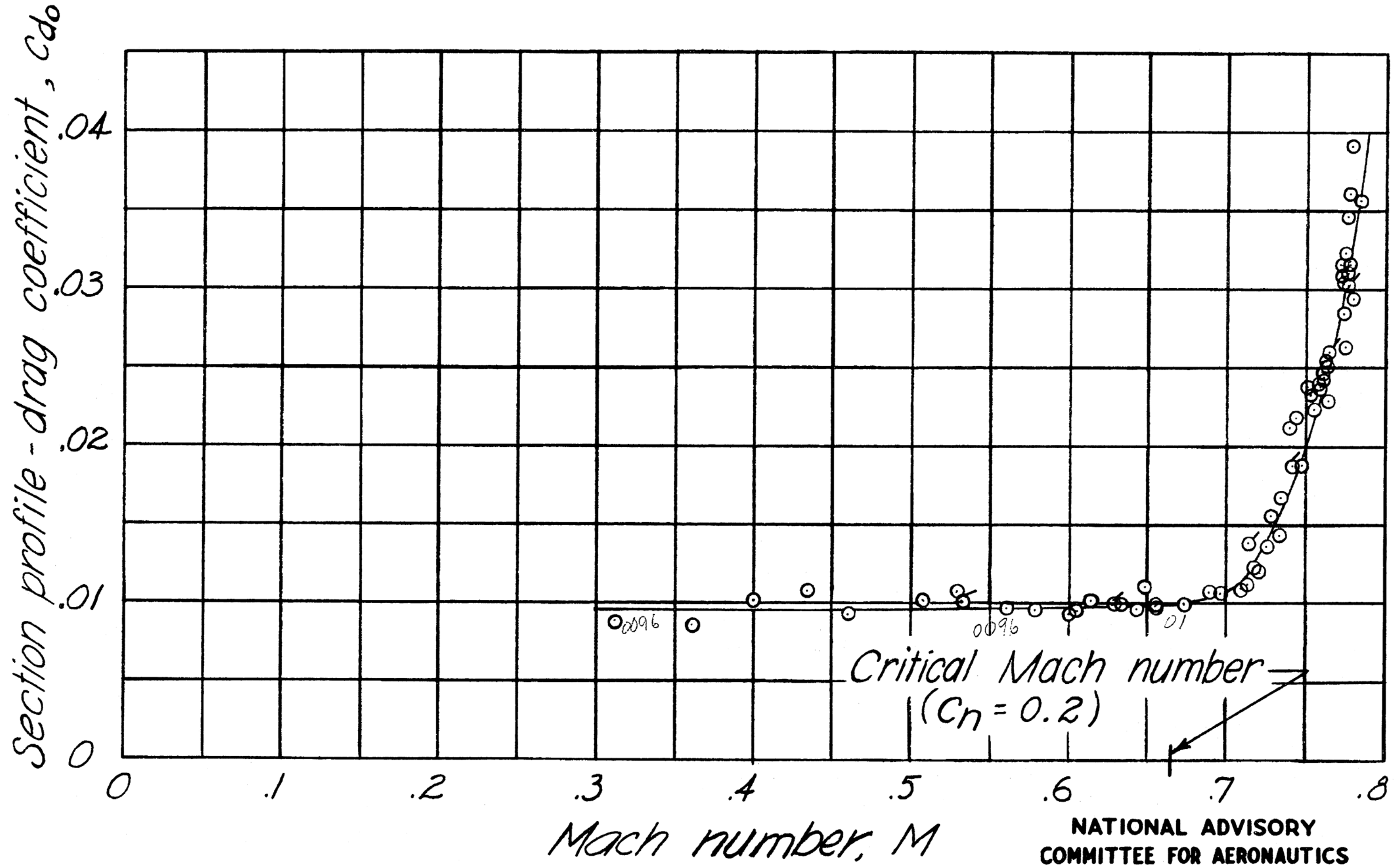


Figure 7.- Variation of section profile-drag coefficient with Mach number at semispan station. $0 < c_n < 0.4$. Symbols with tails attached represent points from figure 5.

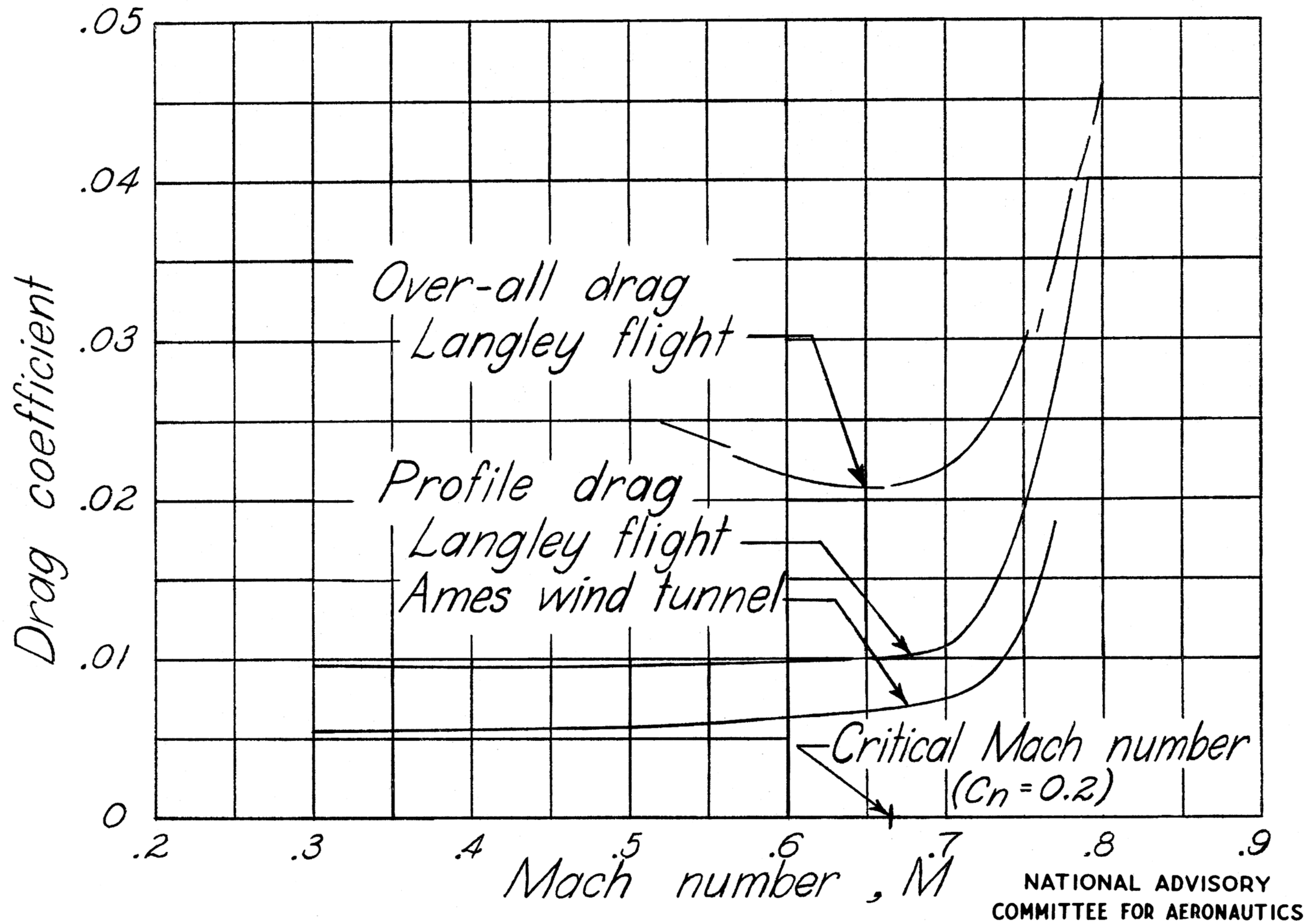


Figure 8.- Comparison of drag coefficients obtained from Langley flight tests with drag coefficients obtained from Ames wind-tunnel tests.

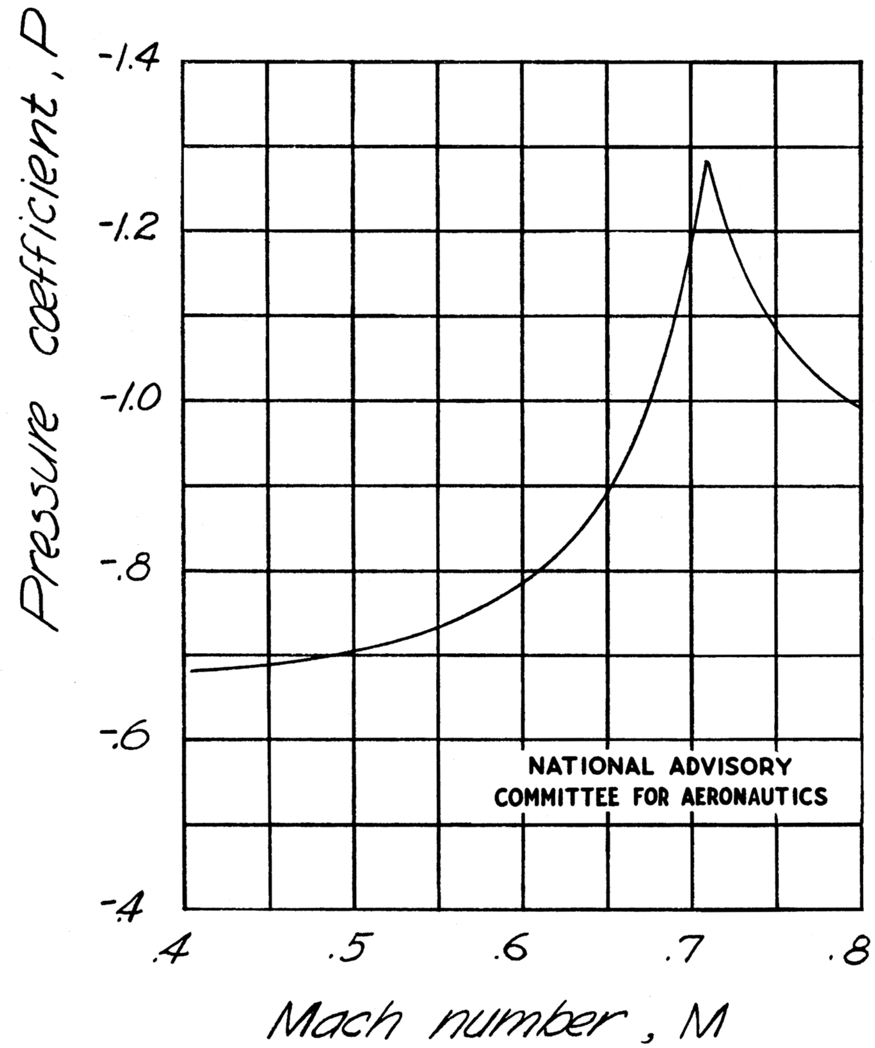
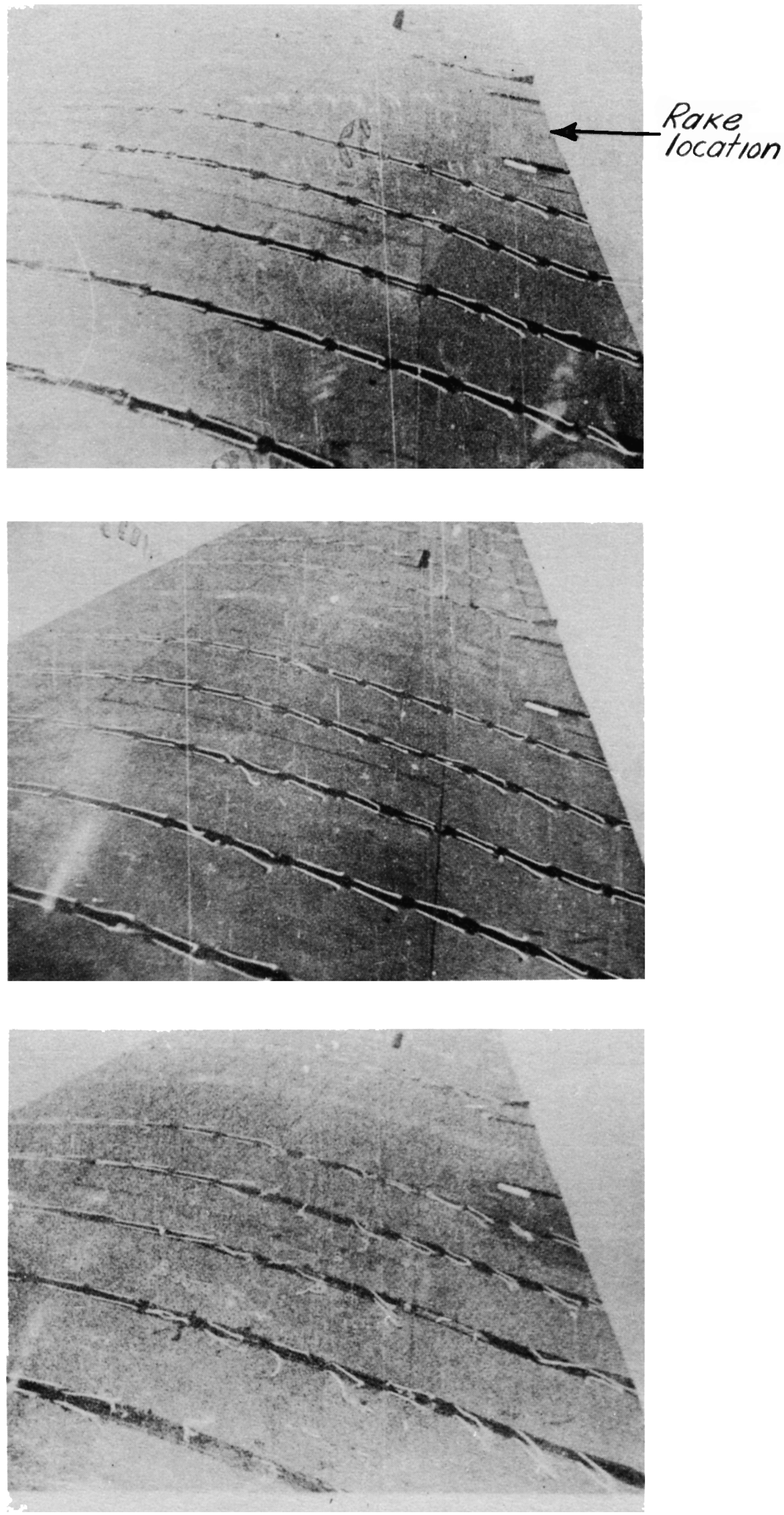


Figure 9.- Local wing-surface pressure variation with Mach number. Station, semispan, 42.2 percent local chord. $c_n \approx 0.2$.



M = 0.65

M = 0.73

M = 0.75

Figure 10.- Flow conditions over upper surface of airplane wing as indicated by wool tufts. Airplane lift coeffiecient, 0.2.